Spacecraft Design Study GROUP AB

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1 Introduction

The purpose of this report is to design a spacecraft mission to Mars, with focus on the power and thermal systems. The goal of the mission is to conduct a photographic survey of the Martian surface using a camera with a resolution of 400x400 pixels, which images a surface area of 200x200 km. The total mission duration is 3 years, taking the spacecraft from lower Earth orbit to a Mars polar orbit, where it will image the surface of the planet. The required thermal environment inside the satellite is -20 to +40 $°C$ during operation, and -50 to +80 $°C$ when the camera is turned off.

2 Mission analysis

The trajectory analysis is conducted by using a Hohmann transfer and a small maneuver for inclination change. The mission is consisted of going from LEO to Sphere of Influence (SOI) of Earth, from Earth's SOI to Mars Transfer Orbit (MTO), from MTO to Mars' SOI and from Mars' SOI to Low Mars orbit (LMO).
The data used for the calculations of the total ΔV needed to reach orbit around Mars are detailed in table 1. In this table, M_x are the masses, r_x are the radius of orbits around the sun, r_{LEO} is the radius of the low Earth orbit, r_{LMO} is the radius of the low Mars orbit, $\mu_X = G * M_x$ and finally $a = \frac{r_E + r_M}{2}$.
After launch, the spacecraft is assumed to be placed on a low circular Earth orbit (LEO) at an altitude of 300 km.

Table 1: Numerical data for the calculations of the ΔV

M_{earth}	M_{sun}	M_{mars}	r_E	r_M	r_{LEO}
$5.975 * 10^{24} kg$	$1.989 * 10^{30} kg$	$6.387 * 10^{23} kg$	$149.5 * 10^9 m$	$227.8 * 10^9 m$	$6.67123 * 10^6 m$

r_{LMO}	G	μ_E	μ_S
$3.932153 * 10^6 m$	$6.67 * 10^{-11} m^3.kg^{-1}.s^{-2}$	$3.985 * 10^{14} m^3.s^{-2}$	$1.326 * 10^{20} m^3.s^{-2}$

μ_M	a
$4.26 * 10^{13} m^3.s^{-2}$	$1.886 * 10^{11} m$

The velocities of the satellite on a circular orbit around a planet is given by equation 1.

$$\boxed{v = \sqrt{\frac{\mu_x}{r_x}}} \tag{1}$$

This equation gives a velocity $v_{LEO} = 7.728 km.s^{-1}$ at 300km and $v_{LMO} = 3.2915 km.s^{-1}$ at 600km for Mars.

An optimization of the Hohmann transfer is used, called Oberth transfer, which allows to reach Mars from Earth [2]. It consists of:

- Thrusting from LEO to transfer orbit to Mars
- Thrusting from transfer orbit to Mars orbit.

Considering that the aim of the launch is Mars observation, a polar orbit at 600 km will allow to cover the whole surface of the planet. A change of orbit plan is then necessary from equatorial plane to polar orbit. This change is assumed to be done during the transfer from Earth to Mars. The ΔV needed for this maneuver is considered very small compared to the ones needed for the other transfers and it is neglected in the following calculations.

<u>Thrust from LEO to MTO</u>

This transfer includes several steps done at the same time, including:

- Thrusting from LEO to Earth's SOI.
- Thrusting from Earth's SOI to MTO.

The velocity of the spacecraft on the transfer orbit at the Earth level is given by equation 2.

$$\boxed{V_{T1} = \sqrt{\frac{2*\mu_S}{r_E} - \frac{\mu_S}{a}}} \tag{2}$$

The velocity of the Earth around the sun is given by equation 3.

$$\boxed{v_{earth} = \sqrt{\frac{\mu_S}{r_E}}} \tag{3}$$

The escape velocity of the spacecraft from LEO is given by equation 4.

$$\boxed{v_{esc_{LEO}} = \sqrt{\frac{2*\mu_E}{r_{LEO}}}} \tag{4}$$

Finally the ΔV_1 necessary to reach the transfer orbit to Mars from LEO is given by equation 5.

$$\boxed{\Delta V_1 = \sqrt{v_{esc_{LEO}}^2 + (V_{T1} - v_{earth})^2} - v_{LEO}} \tag{5}$$

The numerical results for this first ΔV_1 are given in table 2.

Table 2: Numerical results from LEO to transfer orbit.

$v_{esc_{LEO}}$	V_{T1}	V_{Earth}	ΔV_1
$1.093 * 10^4 m.s{-}1$	$3.2723 * 10^4 m.s{-}1$	$2.9781 * 10^4 m.s{-}1$	$3.591 * 10^3 m.s{-}1$

Thrust from MTO to LMO

The ΔV_2 necessary to goes from transfer orbit to the final orbit around Mars can be calculated with the same equations as previously.
The numerical results for this second ΔV_2 are given in table 3.

Table 3: Numerical results from transfer orbit to final orbit around Mars.

$v_{esc_{LMO}}$	V_{T2}	V_{Mars}	ΔV_2
$4.654 * 10^3 m.s{-}1$	$2.1473 * 10^4 m.s{-}1$	$2.4126 * 10^4 m.s{-}1$	$2.0655 * 10^3 m.s{-}1$

In the end this trajectory recquires $\Delta V_{total} = 5.6565 * 10^3 m.s^{-1}$.

Time of travel

Assuming that the satellite reaches Mars in half distance of the elliptical transfer orbit, Equation 6 gives the time of the travel.

$$\boxed{P^2 = \frac{4*\pi^2*a^3}{\mu_S}} \tag{6}$$

The resulting average time of flight is then around 8.6 months.

3 Spacecraft design

3.1 ACDS

The attitude control system (ACS) is intended to stabilize the satellite along its three axis. It relies on 2 star trackers, 4 sun sensors, IMU and 4 Reaction wheels. The ACS secondary objective is to allow the spacecraft to assume different configurations during the phases of the flight, namely the detumbling, the cruise and the orbital phase.
Here it is important to say that the attitude of each payload component has not been taken into account, i.e. the orientation of the gimbal of the solar panels is not included in the ACS, while the influence of external disturbances, like the solar wind hitting the panels, are embodied in the control algorithm.

3.1.1 Assumptions

The requirement for this mission is to have an accuracy around 1 deg. This allows us to use off-the-shelf sensors and to ease orbit prediction, since no hard algorithm, i.e. three-way doppler effect, will be considered. Instead, one-way Doppler effect combined with Kalman filtering will make the estimation strategy for the orbit. Eventually, data from star trackers and sun sensors will be considered enough to satisfy the accuracy requirement.

3.1.2 Sensors

For attitude estimation, 2 star trackers were chosen as the main sensors because of their high accuracy. Star trackers have full autonomy in attitude acquisition and update. However, they are really sensitive to sunlight, from which they can be damaged in case of direct exposure. For this reason, 4 coarse sun sensors are required to improve the robustness of the system.
Two sun sensors will be placed on the same side of the solar panels, so they will be an indicator of sunlight.

An IMU is demanded to keep track of the spacecraft motion. It contains a gyroscope, an accelerometer and a magnetometer. Data will be reconstructed and stored on-board until a communication window will be available, hence they will be downloaded through the telemetry channel. The reconstruction process will end saving attitude information as quaternions. Eventually, the attitude will be included in a Kalman filter in order to estimate and evaluate the orbit propagation.

3.1.3 Attitude control and further problems

Four Reaction wheels are used for attitude control, 3 positioned along the main three axis and one is placed for redundancy. Moreover, eight thrusters for attitude control are used to help the RW both performing angular momentum desaturation (AMD) and maintaining the attitude while exiting the cruise phase.
Three main control modes has been evaluated for this mission: Regular control mode (RCM), namely the normal configuration for detumbling and cruise phase, Image Acquisition Mode (IAM), which is the nominal attitude the satellite shall have during Mars orbit, Breaking Mode (BM), used in mars orbit injection.
The spacecraft during cruise, and during orbit around mars, encounters external torque that can influence its attitude and saturate the Reaction wheels, especially the problems regarding *Solar pressure*, *Outgassing phenomena* and *thruster calibration*. The latter is less intuitive with respect the other two. Actually, the ACS thruster are paired in a way that the total DV is zero. However small misalignment can occur and in-orbit calibration is demanded before starting operations.
AMD will be done weekly when the angular momentum of at least one reaction wheel will reach approximately 60% of maximum value. It will be achieved by means of ACS thrusters.

3.2 Payload

3.2.1 Camera

The camera that is provided for the mission has the following specifications: 5 kg of mass, 5W of power consumption when turned on, and a resolution of 500 m per pixel that covers an area of 200 km x 200 km of the Martian surface. This directly translates to an image size of 400x400 pixels, which roughly has a data size of $50 - 150$ kB.

3.2.2 Image data size

The area covered by the satellite orbiting Mars is a circular sector. At an altitude of 600 km above the surface, the velocity is 3.29 km/s. This velocity gives the angular orbit velocity of $\omega_{mars} = 0.00082 \quad rads/s$. ~~Thus, in 1 second the satellite travels 0.00082 radians.~~ The circle sector is then calculated from $S = \theta * r$, with θ being the angle in radians and r the radius of the circle, meaning the radius of Mars. This gives a value of 2777.7 m. For the camera to cover the entire surface, it must take a picture for every 200 km of the Martian surface. This means the camera takes a picture once every 72 seconds, which is 106.5 pictures per revolution, in case the planet is always in sunlight. With an image size of $50 - 150$ kB per picture, this gives a total data size of $5.33 - 15.98$ MB. This value is considered as the "worst" case scenario that can be encountered during one year, so it will be used to dimension the on board hard disk.

3.3 Power System

The purpose of the power system of the spacecraft is to provide electrical power for the payload and for subsystems supporting it, including transmitter sending images back to Earth. Sufficient amount of power shall be delivered in all conditions during the mission; in transfer orbit from Earth to Mars and in orbit around Mars both in eclipse and in sunlight.
The structure of the power system is shown in figure 1. All the components shown in this graph are listed next and described briefly. Solar arrays and batteries are discussed more thoroughly later in this chapter.

- **Solar array:** Primary power source of the spacecraft. Converts solar irradiance to electric power through photovoltaic effect. Can produce power only when in sunlight.
- **Batteries:** Secondary power source of the spacecraft. Provides power by discharging itself when solar panels are not active.
- **Shunt voltage limiter:** If solar array produces excess power this excess power is discharged through shunt voltage limiter. This maintains steady voltage output from the solar array. [6]
- **Charge control:** During battery charging, a charge controller makes sure that the battery is not overcharged and that the charging happens in tolerable current level
- **Discharge by pass:** If there is a need to discharge the battery for maintenance purposes it is done through discharge by pass in a controlled manner with a current rate that does not harm the battery
- **Regulation:** Regulates the voltage supplied for the load keeping the voltage at steady constant level.

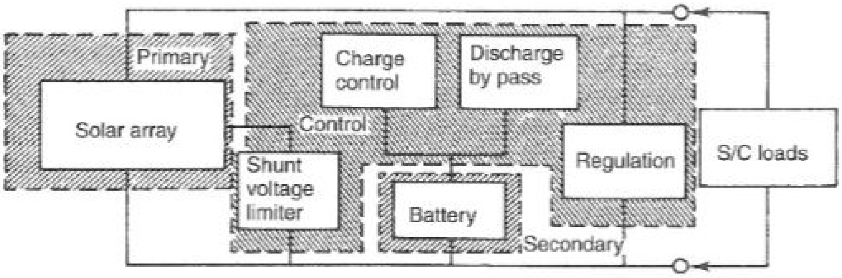

Figure 1: Diagram of the spacecraft power system [5]

3.3.1 Solar Panels

In this chapter the the design process of the solar panels is discussed. The chapter begins with study regarding the eclipse times which are needed for secondary power system (batteries) dimensioning. After this dimensioning of the panels so that they can produce sufficient amount of power for the spacecraft and factors affecting to this dimensioning are introduced.
During the mission, the main power source for instruments on-board the spacecraft are solar panels. The power produced depends on irradiance hitting the solar panels. This energy flux is not constant during the mission. Solar irradiance depends on the distance from the Sun and on the fact that the spacecraft can be in shade of a planet or other object. In this report it is assumed that the radiative power of the Sun follows a long term average value and remains constant. The dimensioning will be taken when the satellite is orbiting Mars, since the irradiance will be less than LEO.

During the cruise phase,the spacecraft travels further from the Sun and the Solar irradiance decreases. Here, Sun is considered as point-source radiation. This means that energy is inversely proportional to the square of the distance from the Sun. The average solar power flux at Earth orbit and Mars are known to be $I_{Earth} = 1361.0W/m^2$ [1] and $I_{Mars} = 586.2W/m^2$ [3]. Hence, the Solar irradiance hitting the solar panels of the spacecraft during the cruise phase at distance x from the Sun can be calculated with equation 7.

$$I_x = I_{Earth}(\frac{r_{Earth}}{x})^2 \tag{7}$$

Where r_{Earth} is the average radius of Earth's orbit around the Sun which is $149.60 * 10^6 km$. The variation of the solar irradiance during the mission is shown in figure 2 as a function of distance from the Sun.

When the spacecraft has started to orbit Mars in circular polar orbit the spacecraft is from time to time in nightside and in dayside. When Mars blocks the sunlight and the spacecraft is in eclipse solar panels cannot produce power and instruments on board the spacecraft must rely on batteries. When the spacecraft is not behind the planet the solar panels receive a solar irradiance with an average value of $586.2W/m^2$ (this value depends on the fact that Mars is closer to its perihelion or aphelion. In this report this dependency is neglected).

The time spacecraft spends in sunlight and in eclipse during one orbit around the planet depends on the angle between sun rays and the orbital plane (figure 3). It can be assumed that the orbital plane of the spacecraft remains parallel with constant direction in space. This means that as the Martian year proceeds the angle between Sun's radial axis pointing towards Mars and orbital plane changes. This leads into change of the time the spacecraft spends in eclipse during one orbit around the planet. From figure 3 it can be seen that when the mentioned angle is 0° the time spacecraft spends in eclipse is the longest. When the mentioned angle is 90° the spacecraft does not enter the the eclipse at all during its orbital period. Figure 4 shows the durations the spacecraft spends in sunlight and in eclipse in different parts of the mission during one orbit around Mars. From the mentioned figure it can be

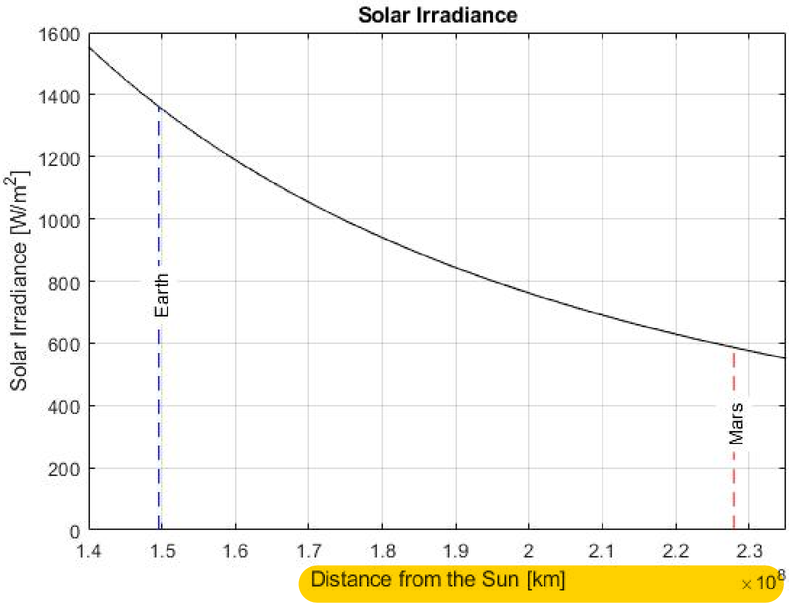

Figure 2: Solar irradiance during the mission as a function of distance from the Sun

seen that the longest continuous time spacecraft would spend in eclipse would be $41min$. It can also be observed that the spacecraft has periods when it does not enter the eclipse at all during its orbit. These periods last 0.32 Earth years and happen every 0.94 Earth years.

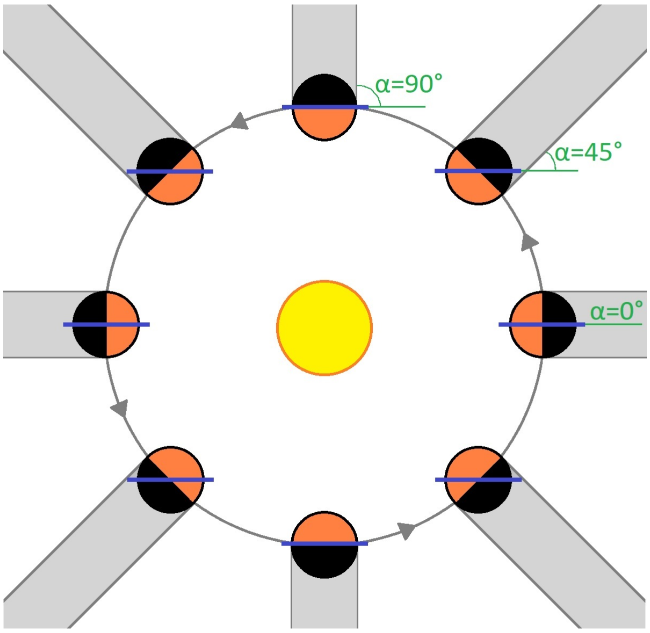

Figure 3: Sketch showing how the angle between sun rays and the orbital plane changes during a Martian year. Sketch shows Mars in 8 different positions around the Sun during a Martian year. Blue line represents the orbit of the spacecraft around Mars.

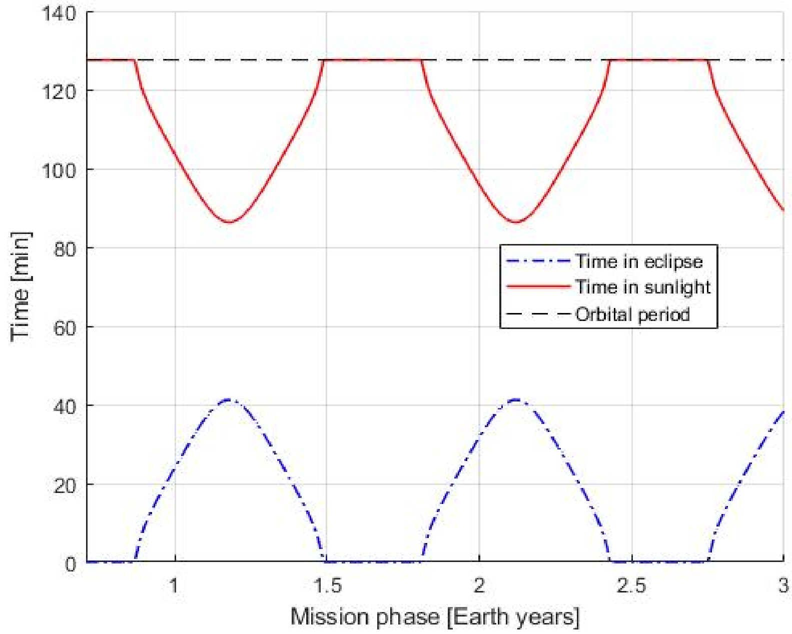

Figure 4: Time span the spacecraft would spend in eclipse and in sunlight during single orbit around Mars in different phases of the mission.

The following assumptions were made during the study of eclipse ~~duration to make the calculations simpler.~~

- Mars trajectory is assumed to be circular
- Orbital plane of the spacecraft is perpendicular to Mars orbit around the Sun. This deviates few degrees from the polar orbit of the spacecraft that it would really travel
- Time of the spacecraft spends in Mars' penumbra is considered to be insignificantly small
- Border of Mars' umbra is considered to be parallel with a line pointing from Sun to Mars

When the maximum eclipse time and seasons where the spacecraft would be all the time in sunlight were determined, the battery system forming the secondary power source could be designed. Then the power needed for charging the batteries could be defined which sets the limits for solar panel design. The solar cell used is a "Triple Junction Solar Cell 3G30C-Advanced (BOL)" manufactured by Azur space [8]. The characteristics of this cell are given by the manufacturer and are presented in table 4.

Table 4: Characteristics for "Triple Junction Solar Cell 3G30C-Advanced (BOL)" solar cell when irradiance is $1367 W/m^2$ and temperature is $28°C$ [8]

Physical	
Width $[mm]$	40.15
Length $[mm]$	80.15
Thickness $[mm]$	0.28
Absorptivity coefficient	≤ 0.91
Electrical	
Voltage at max power $[V]$	2.409
Current at max power $[A]$	0.5029
Average efficiency	0.293
Power degradation $[\%/year]$	5
Temperature gradients	
Voltage at max power $[V/°C]$	-0.0067
Current at max power $[A/°C]$	+0.00024

The solar panel configuration is designed assuming the worst case conditions that can be present during the mission. This means that for the design condition the degradation of the solar cells will have continued the longest possible time and the spacecraft will be at its furthers point from the Sun it will achieve during the mission. In these conditions the solar panel efficiency is at its lowest point and the magnitude of the solar irradiance reaching the panel is at minimum. These conditions are present when the spacecraft is just ending its three year mission and it is orbiting Mars. The characteristics of the solar panel design are presented in table 5.

Table 5 also describes how the solar panel performs in different phases of the mission. When the spacecraft starts the mission and is near Earth, solar panels produce much more power than near Mars where the solar irradiance is considerably smaller. From the table it can be seen that the power produced is larger when the spacecraft arrives to Mars than what it is when the mission ends. This difference is caused by power degradation due to radiative effects on the solar cells. One thing worth noting is that solar panel produces a considerable amount of excess power during every part of the mission. This allows some divergence of the panel from the perpendicular orientation in relation to the Sun rays resulting to some drop in produced power. This means that solar panels do not need to face Sun in perfectly perpendicular angle and attitude corrections are not needed that frequently. Excess power also makes the power system to tolerate a loss of few solar cells and other unexpected malfunctions on-board which can result in an increase need of power.

Table 5: Characteristics of the spacecraft solar panel in different phases of the mission

Configuration properties	Leaving LEO	Arriving to Mars	Mission end
Cells in series	13	Same	Same
Cells in parallel	108	Same	Same
Cell amount	1404	Same	Same
Mass $[kg]$	5.34	Same	Same
Area $[m^2]$	4.52	Same	Same
Environment conditions			
Temperature $[°C]$	$28°C$	$28°C$	$28°C$
Solar irradiance $[W/m^2]$	1367.0	588.9	588.9
Electrical properties			
Efficiency	0.293	0.283	0.250
Supply voltage $[V]$	31.3	30.8	28.9
Supply current $[A]$	54.3	23	21.58
Supply power $[W]$	1700.9	706.8	622.9
Excess power $[W]$	1100.9	106.8	22.8
Power per panel unit mass $[W/kg]$	319.0	132.6	116.9

The following assumptions were made during the design study of solar panel.

- Degradation happens at constant rate during the entire mission
- Power can be drawn from the solar panel so that the solar cells can be operated at their maximum power point. This means that the shunt voltage limiter is assumed to be capable of consuming the excess power from the solar panel at all conditions and the voltage regulation makes sure that the voltage of the main bus remains at relatively steady value of $28V$
- A control on the gimble makes sure the solar panels are facing the Sun constantly with a decided angle from ground control.
- All the solar cells produce power simultaneously, meaning that there is no mismatch between them.

3.3.2 Batteries

Batteries are intended to provide power when solar panels are not in use, that is during launch and eclipses around Mars. The scaling of the battery will only take into account the power needed during the eclipses. It can be assumed that the satellite will be hit by sunlight all along its cruise between earth and mars. The solar panels will provide the required power during this phase. During the eclipses around mars, batteries will take over before the satellite reaches sunlight again. They will then be recharged. During the eclipse time, the satellite will be put on stand-by mode as telemetry and imaging can't be used. However, some power will be required mainly for the ACS, the calibration of the camera and other subsystems such as the on board computer, active thermal control. The continuous maximum power required during the eclipse time is fixed at 50 Watts in a conservative way, which represents roughly the total energy of 35 Wh for the longest eclipse duration of 41 minutes. This energy will mainly be used occasionally during the attitude control maneuvers.

The batteries chosen for this mission are cylindrical lithium-ion cells, known for their high specific energy, in this case 154 Wh/Kg. The charging of those batteries requires a suitable regulation of the voltage and current delivered from the solar panels. A charge circuit is provided by the manufacturer, above the battery pack. The main characteristics of the Li-ion cells chosen are summed up in table 6. Those data are inspired by GomSpace Li-ion 1850 battery pack, used for space applications [10].

Table 6: GomSpace Lithium-ion cell 18650 main characteristics

BATTERY CELL	
Battery type	Li-ion
Cell voltage (V)	3.7
Cell capacity (Ah)	2.6
Energy (Wh)	9.6
Height (mm)	65
Diameter (mm)	18.5
Volume (cm^3)	17.5
Weight (g)	48

The chosen voltage at the battery is 29.6 V, a common used voltage for space applications. The battery packs will contain 8 cells in series, according to the 8S-1P configuration from the manufacturer [11]. The required capacity depends on the lifetime needed for the mission. Assuming that the satellite will orbit during 2.3 years around mars, the batteries will have to make a maximum of 6242 cycles, discharging during a time comprised between a few seconds and 41 minutes. For safety reasons, all the cycles are assumed to last 41 minutes. By choosing a low Depth Of Discharge, the lifetime of the battery is extended. According to the manufacturer's data [10], a DOD of 6.8% ensures a long enough lifetime, assuming that dividing the DOD by 2 leads to a 2 times longer lifetime. The capacity loss is given at 5% per year. Thus, the required battery pack characteristics are presented in table 7.

Table 7: Battery packs characteristics

BATTERY PACKS	
Pack configuration	8S-1P
Voltage (V)	29.6
Capacity (Ah)	20.8
Number of cells in series	8
Number of cells parallel	8
Number of battery packs	8
Total number of cells	64
Total volume (dm^3)	2.623
Total weight (kg)	4
Power dissipation (W)	4.48

The maximum charging time at 0.1C is estimated at 2000 seconds with a DOD of 6.8%. The battery have enough time to recharge even during the shortest light period. The voltage should remain between 3 and 4.2 V for each cell. During charging, the temperature of the batteries should remain between 0 and 45 degrees Celsius and between -20 and 60 degrees Celsius when discharging.

3.4 Propulsion system

The Mass Orbiter Mission (MOM) is considered for inspiration in the propulsion system choice due to use of the same payload [7]. Two propulsion systems are used for this mission. One main engine of 440N thrust to send the satellite from Earth to Mars and eight small thrusters of 22N for attitude control. The propellant used is a bi-propellant consisting of monomethylhydrazine (MMH) and dinitrogen tetroxide (N_2 O_4).

The rocket equation is used to estimate the mass of the propellant required.

$$\Delta v = I_{sp} * g * \ln \frac{m_{wet}}{m_{dry}} \tag{8}$$

where,

$$m_{wet} = m_{dry} + m_{prop} \tag{9}$$

The mass of the propellant is calculated around 960kg. As a safety factor, a total mass of about 1-1.1 tonnes of propellant is used to account for attitude control maneuvers. Two pressurized tanks are used, one for the propellant and one for the oxidizer, with an estimated volume of 400L and a mass of 35kg each, similar to MOM mission. Finally, the dry mass of the satellite is estimated about 185kg.

4 Thermal System

4.1 Thermal model

Equipment have been placed on the faces of the satellite according to the requirements of the mission. Their places are detailed in table 8. Only the solar panels have been modeled on Thermica. -Y is pointing to Mars and the velocity of the satellite is parallel to +Z. Radiators will be placed on -Y, +Y, -Z and +Z.

Table 8: Place of the equipment on the different surfaces of the satellite.

Face	+X	-X	+Y	-Y	+Z	-Z
Equipment	Solar panel	Thruster	Telemetry	Camera	Part of the tanks	/

The dimensions of the satellite are detailed in table 9.

Table 9: Dimensions of the satellite

	Cube	Solar panels
Width	1.5 m	1 m
Length	1.5 m	4 m

In this simulation, the spacecraft is considered in its orbit around Mars in two different thermal cases which represent the extreme conditions that it will encounter. Considering its elliptical orbit, Mars is closer to the Sun during the nothern winter than in summer. It means that the two cases are:

- **Cold case:** summer solstice in begining of life (BOL). 20 november in 2017 [13].
- **Hot case:** winter solstice in end of life (EOL). 16 october 2018 [13].

4.2 Net thermal flux

The absorbed flux of each surface M_a can be calculated with equation 10

$$\boxed{M_a = M_{sun} + M_{MAlbedo} + M_{MIR}} \tag{10}$$

where M_{sun} is the absorbed flux due to the sun thermal flux, $M_{MAlbedo}$ due to the Mars albedo and M_{MIR} due to the Mars IR. These three absorbed flux have been calculated on Thermica and are shown in tables 10, 14.

Table 10: Absorbed flux in cold cas (W/m^2).

Face	M_{sun}	$M_{MAlbedo}$	M_{MIR}	M_a
+Y	10.1	0	0	10.1
-Y	10.1	1,2	66.5	77.8
+Z	10.1	0.3	16.8	27.2
-Z	10.1	0.3	16.7	27.1

Assuming that the radiators can be considered as gray bodies the emitted flux of a surface can be calculated with equation 11.

Table 11: Absorbed flux in hot case (W/m^2).

Face	M_{sun}	$M_{MAlbedo}$	M_{MIR}	M_a
+Y	15.2	0	0	15.2
-Y	15.1	1.8	70.7	87.6
+Z	15.1	0.5	17.8	33.4
-Z	15.2	0.4	17.7	33.3

$$\boxed{M_e = \epsilon * \sigma * T^4} \tag{11}$$

where T is the temperature of the radiator in kelvin and σ is the Stefan Boltzmann's constant. T is considered to be 0° in the cold case and 20° in the hot case. In the beginning, the satellite platform is considered to be made of SSM Aluminum whose emissivity coefficient is 0.78 and absorption coefficient is 0.15 at BOL and 0.19 at EOL.

In the end, the net thermal flux per unit surface is given by equation 12.

$$\boxed{M = M_e - M_a} \tag{12}$$

The calculations give the results in table 12. Considering these results ADCS is placed on +Y, On Board Computer (OBC) is placed in -Y and the batteries are placed on +Z.

Table 12: Net thermal flux per unit surface (W/m^2).

Face	+Y	-Y	+Z	-Z
Net Flux Cold Case (W/m^2)	236	168	218	219
Net Flux Hot Case (W/m^2)	311	238	293	293

The final configuration of the equipment is then the one described in table 13.

Table 13: Final places of the equipment on the different surfaces of the satellite, masses and dissipated powers.

Face	+Y	-Y	+Z	-Z
Equipment	Telemetry+ADCS	Camera	Tank+OBC	Batteries
Hot case dissipation (W)	18.2 + 112	3.5	15	4.5
Cold case dissipation (W)	0 + 50	3.5	10	1
Mass (kg)	4.9 + 30	5	5	4

4.3 Radiators

Assuming that the radiators temperature should remain below 20°, the area of the radiators can be determined by writing the thermal equilibrium as in equation 13.

$$\boxed{A = \frac{P_d}{\epsilon * \sigma * T^4 - M_a}} \tag{13}$$

where P_d is the power dissipated.

Using the values for the hot case of table 14 and table **??**, the radiators should have the areas written in table **??**.

Table 14: Minimum areas for the radiators in order to keep the temperature below 20°.

Face	+Y	-Y	+Z	-Z
Radiator area (m^2)	0.42	0.015	0.051	0.015

4.4 Heaters

In order to keep the temperature upon 0°, heaters can be placed on each face. Their power can be determined with equation 14.

$$\boxed{M_H = A * \epsilon * \sigma * T^4 - A * M_a - P_d} \tag{14}$$

where M_H is the power of the heater, A the area of the radiator.

With $T = 0°$ in the cold case and using the values for the cold case of table 14 and table 13, the heaters should have the power written in table 15.

Table 15: Minimum power for the heaters in order to keep the temperature upon 0°.

Face	+Y	-Y	+Z	-Z
Heater power (W)	47	-1.2 *	0.88	2.3

* Considering the minus sign, -Y face will not need any heater.

4.5 Results

Table 16: Temperatures of the satellite faces with radiators and heaters.

Face	+Y	-Y	+Z	-Z
Temparature in the hot case				
Temparature in the cold case				

5 Conclusions

The feasibility study of a mission to Mars has been evaluated. More efforts have been put to Power system. A ~~summary of relevant data is provided~~ in Table 17.
In order to start with numerical data, two missions to Mars that occurred in the past, have been investigated. One from NASA called Mars Recoinnaissance Orbiter (MRO [4] and another from Indian space agency, the mission was called Mars Orbiter Mission (MOM [7]). Also some data have been inspired by Trace gas Orbiter (TGO [9]) mission from ESA.
~~Eventually,~~ the data which have been sorted out for the design of this spacecraft are consistent with the previous missions.

Table 17: Table with mass and power consumption for each subsystem

Subsystem	Mass [Kg]	Power [W]
ADCS	30	450
Telemetry	4.9	
Camera	5	5
OBC	5	
Batteries	4	50

6 Division of work

A. Cisowski Worked on the Mission Analysis

R. Farid Worked on the Payload

F.Giuliano Worked on ADCS and collaborated to power system. He also wrote about conclusions. Helped in reviewing.

J. Imbert Worked on the Thermal System: estimation of radiators, heaters and satellite temperatures. Worked on the Mission Analysis.

Y. Jiao Worked on thermal analysis

C. Muresan Worked on telemetry

M. Nicolle Was responsible for the study related to batteries and their scaling

K. Papavramidis Worked on the Mission Analysis, Propulsion System and Power system. Responsible of reviewing everything.

S. Simolin Was responsible of the study related to solar panels and their dimensioning

F. Raiti Choice of attitude determination sensors, estimation of reaction wheel power consumption, thermal simulation and estimation of radiator areas and heater powers.

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